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VENUS AND BEYOND USING THE ARIANE ASAP LAUNCH CAPABILITY*

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A general trajectory scenario has been designed which can transfer a spacecraft from a pre-specified, highly eccentric, near equatorial orbit to a desired escape direction. The spacecraft requires a restartable engine and must perform close flybys of the Earth and Moon. This process will be used in 2003 by NASA to launch small spacecraft to Mars piggyback on the Ariane 5, departing from its Kourou launch site in French Guiana. The small spacecraft will be released from the upper stage of the Ariane into the geosynchronous transfer orbit (GTO), which is used to deliver the comsats to GEO. Once released into GTO, the spacecraft must remain in this, or higher orbit, for perhaps months, until about 2 months before Earth escape. At this time, the transfer process begins with the spacecraft firing its engine to place it at a point beyond the Moon's orbit, followed by a second burn to target a lunar flyby, which will direct it to return to an Earth flyby, where a third burn results in Earth escape. The time from the first to the third burn is about 60 days. These burns are planned to minimize the sum of the velocities required to transfer from the given GTO to the proper escape direction. The method has been applied here to Venus missions in the years 2004 to 2009. In addition, since Venus is very effective in the application of gravity assist, examples of its use in reaching other bodies of the solar system are also presented.

INTRODUCTION

The planet Venus is about as accessible, from a launch energy standpoint, as Mars. Almost 2 dozen missions have been flown there, the most successful being Magellan, whose radar instrument penetrated the opaque atmosphere and mapped the surface. Future missions to Venus have been proposed, in NASA's Discovery Program for example, but being cost constrained, a dedicated launch vehicle would consume a significant fraction of the total cost. This cost could be reduced considerably by flying as a secondary payload, but then the question remains as to how the spacecraft is to transfer from the Earth orbit it is dropped into, to the proper escape direction required to get to Venus. It is assumed here that the spacecraft is of low mass, and that the spacecraft and instrument construction has taken advantage of new technology in miniaturization and materials.

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The means proposed here for the orbit-to-escape transfer applies to the piggyback capability provided on the Ariane 5 during it's geosynchronous Earth orbit (GEO) launches. An important benefit of the GTO is that escape from this orbit requires over 2000 m/s less than escape from LEO. Also, the Ariane capacity is substantial. Up to eight 100 kg (or four 200 kg) auxiliary payloads may be carried into the geosychronous transfer orbit (GTO) and released. The large communication satellites are released at GEO altitude and have their own propulsion system to circularize. The secondary payloads are attached to a ring in the upper stage of the Ariane called the Ariane Structure for Auxiliary Payloads (ASAP)[1] and, when released, must have their own restartable propulsion system to depart for Venus from the highly eccentric GTO. The suggestion that this would be a worthwhile problem to investigate was conveyed to me by Blamont of CNES[2] in 1997. A first attempt at solving this problem concentrated on single burns, but hinted at possible lunar gravity assist.[3]

MEGA GENERAL DESCRIPTION

The method of transfer suggested here is called the Moon-Earth Gravity Assist (MEGA) and requires a minimum of three propulsive maneuvers, together with flybys of the Moon and Earth, resulting in proper escape.[4] It takes about 60 days to execute, so that the actual Ariane launch would have occcured at least 60 days before Earth escape and, as will be seen, sometimes months before. As shown in Figure 1, the sequence begins with the spacecraft in the near equatorial GTO orbit, or in some intermediate orbit (1). The first burn (2), performed at perigee and in the orbit plane, transports the spacecraft to some distance beyond the Moon's orbit. At the apogee of this high ellipse (3), a second burn targets to a lunar flyby (4) such that the spacecraft will return

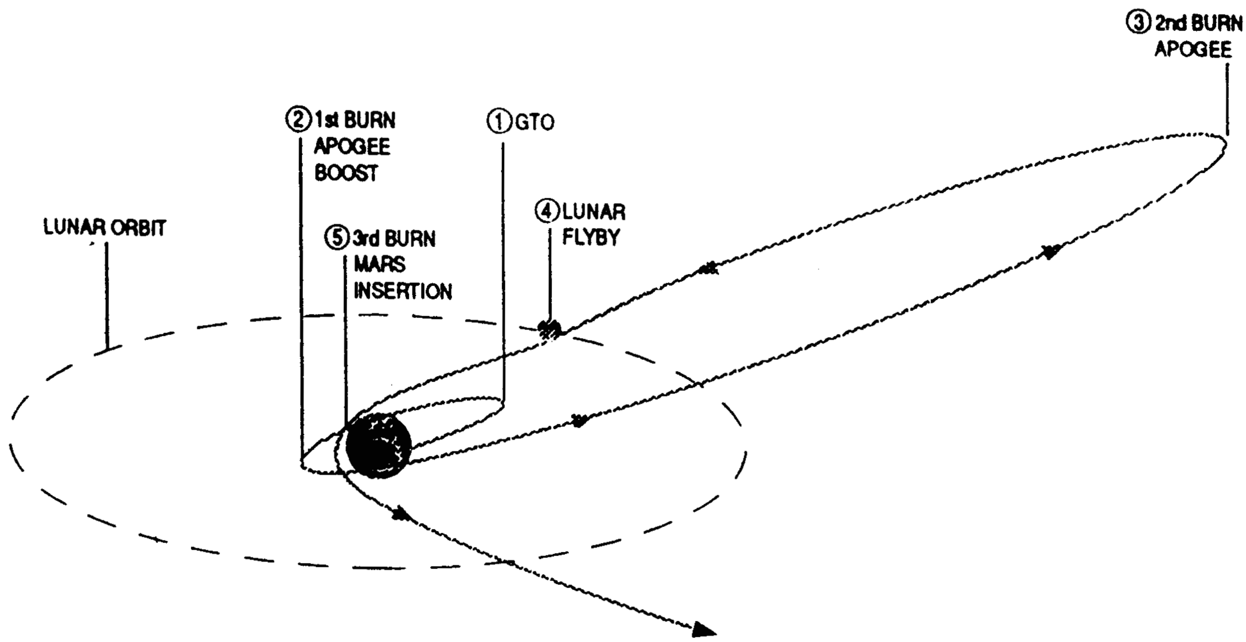


Fig. 1 GTO to Venus Using the 3-Burn Moon-Earth Gravity Assist

to Earth with a low altitude, e.g., 300 km, where a third burn (5) will propel the spacecraft to the required escape direction (and with the necessary velocity) to escape Earth and be on its way to Venus. The Venus craft must have an engine restart capability, and a total velocity capability of at least 1500 m/s to perform the three maneuvers and allow a three-month or more Ariane launch period. Application of this process cannot be computed based on simple conic motion alone. Nodal regression of the GTO, solar perturbation on the high ellipse, and orientation of the GTO axis with the desired escape vector must be taken into account.

MEGA TRAJECTORY BASICS

The GTO is a highly eccentric orbit (250 by 35800 km altitude) and, for the Ariane launch site, is inclined about 7 deg with the Earth's equator. For efficient Earth escape, a single burn only would be performed near perigee and in the plane of the GTO. To do it other than near perigee is not as efficient (it would require a higher ΔV) for a given escape energy. Also, since the GTO is near equatorial, it would be costly to achieve an escape vector with a significant latitude, say, 20 deg. The MEGA method alleviates both of these concerns by placing the perigee of the third burn at the ideal location, so that an in-plane burn there will place the spacecraft on the desired path to Venus. The MEGA process is necessary since, for secondary payloads, no requirements may be placed on the Ariane launch date or time of day, so that the resulting GTO may have any orientation, placing a severe restriction on single burn possibilities. MEGA uses three or more burns to overcome the unfavorable GTO orientation relative to the escape asymptote, while performing the major burns (one and three in Figure 1) in-plane and at a low Earth altitude perigee.

The approach in developing an acceptable trajectory using MEGA requires first selecting a specific mission to fly, say, a Venus launch in 2004, and computing the interplanetary trajectories best suited for the mission's goals, over a period of about a months worth of launch (escape) dates. If maximizing launch mass is the goal (i.e., minimizing escape energy), then the launch dates should be in April of 2004, using a Type 2 transfer to Venus. This results in an acceptable range of escape energies (C_3), and a range of escape directions, in equatorial latitude and longitude, over a period of about a month.

Next, the Moon's location and date, for its flyby on return from the high apogee, can now be selected so that the third, Earth escape, burn will occur at an Earth perigee of 300 km, and in the trajectory plane. In the conic world, the solution will be an elliptic segment going from the Moon to perigee connected to a hyperbolic segment going from perigee to escape. Experience has shown that the Moon's position vector should be about 45 deg ahead of the escape vector in longitude. A table look-up in a lunar ephemeris can then place the date of the lunar flyby fairly accurately. It would be a particular day in April, say, for the 2004 mission. These date approximations, together with an estimate of the set of GTO launch dates, will be used in a computer program to initiate the MEGA calculations.

To this point, we have not been concerned with the lunar flyby itself, but only the Moon's position vector. The flyby conditions at the Moon of altitude and inclination provide the degrees of freedom to match the 300 km perigee at

Earth, and the orientation of the plane containing the outgoing escape vector. Note that, because of this, the approach to the Moon from the high apogee location, where the second burn is done, is of no great concern, as long as the resulting lunar flyby can produce the required Moon-to-Earth ellipse without a lunar impact. In doing the calculation, an iteration is required to match the Moon relative velocity (speed only) leaving the Moon with the lunar approach velocity coming from the high apogee.

The next aspect to be looked at is the required orientation of the GTO relative to the geometry just developed. The first burn is constrained to be done at GTO perigee. Thus, the important element of the GTO is the orientation of the major axis. The apogee location beyond the Moon, where the second burn is made, must be (within limits) on the same side as the Moon's location. To relate the GTO major axis to launch date, we assume that launch is such that the GTO axis points to the sun on the day of launch, which would be approximately noon local time. (Other orientations will be considered later.) Now, the GTO axis angle can be specified in the Ecliptic vernal equinox coordinate system, as shown in Figure 2.

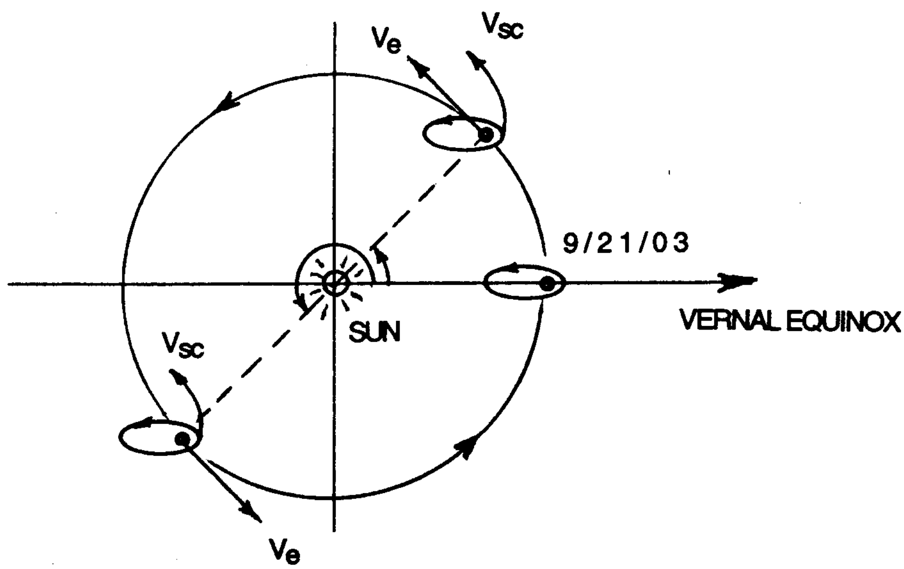


Fig. 2 MEGA Delay periods for Inbound and Outbound Missions

In this Figure, assume that the Ariane launch is on September 21. Then the GTO major axis points to the sun at 180 deg. For Mars, the ideal time for launch is about 2 months later, or in November, because the third burn, done approximately on the same side as GTO perigee, as MEGA requires, would direct the escape in the direction of Earth's motion, good for an outbound mission, say to Mars or Jupiter.

To counter the Earth's motion, as required for Venus or Mercury, the third burn should be done in a direction opposite the Earth's motion, or in May, eight months later, as shown in Figure 2. For the Venus 2004 case above, for example, since the lunar flyby and Earth escape dates are in April, the Ariane launch period would be eight months before, or August 2003. It will be seen that the MEGA process is quite robust, and that the Ariane launch can range from June through October of 2003.

THE ARIANE LAUNCH HOUR

The analysis above assumes a noon apogee arrival which, for a given day, fixes this ellipse in space. However, on the Ariane, secondary payloads will not be able to specify this condition, which will depend on the launch time-of-day. It is therefore necessary to consider other apogee arrival times, specifically, those between 9 am and 6 pm. The MEGA process, then, must apply to this parameter, which we will refer to as hours of apogee arrival referenced from Noon Local Time (NLT), which can range from -12 hr to 12 hr for a given day.

The apogee arrival time in NLT is an input to the MEGA program, so that 3-burn trajectories can be computed for other than noon time arrivals. However, for a given GTO orientation, there is a relationship between day of launch and time of apogee arrival as shown in Figure 3. Here, a give apogee orientation, say a noon time (0 hr) arrival, is equivalent to a 10 am arrival if the launch date is 30 days later, and to a 2 pm arrival if the launch date is 30 days earlier. That is, for 2 pm, for example, with the Earth rotating counterclockwise, the apogee has passed noon by 30 degrees. This equivalence relationship may be used to fill in the launch day launch hour space for the required MEGA trajectories. The same MEGA solution may be used for these three launch dates only if initiation of the 3-burn is on April 18th.

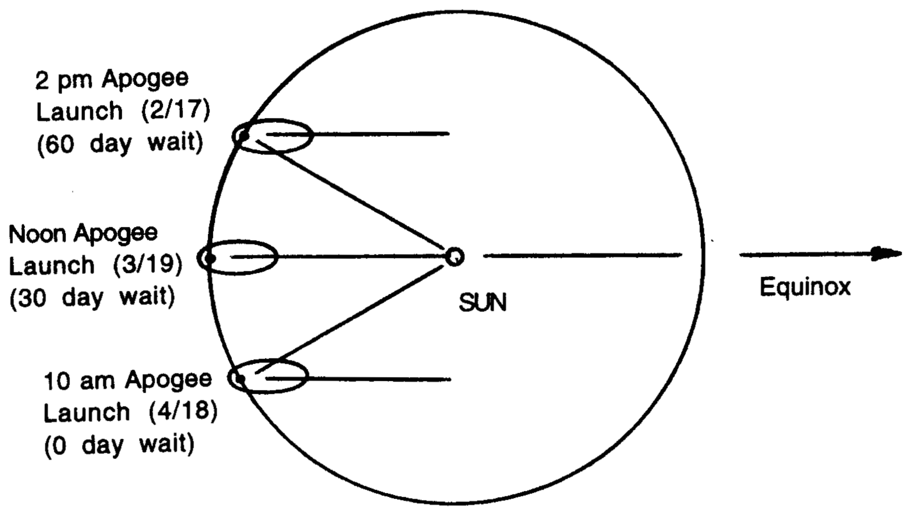


Fig. 3 Launch Day/Launch Hour Equivalence for the MEGA 3-Burn

MEGA OPPORTUNITIES FOR VENUS MISSIONS

Using the MEGA process, Ariane launch periods have been developed for minimum energy transfers to Venus for the years 2004 to 2009. That is, for each Venus opportunity, about every two years, a single trajectory (except for 2005 with two) has been selected for specific days of Earth departure and Venus arrival. For this trajectory, the Venus spacecraft may be on board the Ariane as a secondary payload any time during a 5-month interval. This interval begins approximately 10 months before Earth escape, and ends about 5 months before.

Then, two months before the escape date, the MEGA transfer begins with its first burn into the 60 day high ellipse beyond the Moon's orbit. The duration, or wait time, between the Ariane launch and initiation of the 3-burn (a period of 8 down to 3 months) will probably require the spacecraft to leave the GTO because of the radiation hazard, but this phase of the mission has not been studied.

Table 1 presents the characteristics of the trajectory chosen for each of the launch opportunities, and the bounds on the total 3-burn velocity requirement. These bounds were chosen from the velocity plots shown in Figures 4 to 7.

Going through Table 1, the Earth to Venus type (column 1) refers to whether the heliocentric transfer angle is less than 180 deg (type 1) or greater (type 2). The escape latitude and longitude are referenced to the Earth's equinox and mean equator of date. It can be noted that the escape longitudes are indeed opposite Earth's velocity vector, as required for Venus missions. At Venus, if aerocapture is not used, then the 8th column gives the ΔV at 300 km altitude required to get into a 24 hour elliptic orbit. These may be lower if other than minimum C3 trajectories were chosen to Venus. Finally, the Venus equatorial latitude and longitude of the approach vector is useful in determining minimum inclination orbits and accessible entry or landing latitudes.

Table 1. MEGA Launch and Arrival Conditions for Venus 2004-2009

Launch Year (Type)	Earth* Escape Date	Escape Energy (C_3)	MEGA ΔV Min (m/s)	MEGA ΔV Max (m/s)	Escape Lat.** (deg)	Escape Lon.** (deg)	Venus ΔV 24 hr Orbit	Venus Lat.** (deg)
2004/2	Apr. 4	9.0	1190	1400	38.0	95.8	1860	-29.4
2005/2	Oct. 18	8.7	1110	1290	-18.0	322.6	1480	27.5
2005/2	Nov. 14	8.8	1225	1420	-26.7	304.1	1440	32.4
2007/2	May 30	7.5	1170	1340	-9.2	145.9	1050	-12.0
2009/1	Jan. 9	8.0	1200	1380	11.2	8.8	1230	45.1

These values apply over a 5-month launch period.
*The lunar flyby is 2-4 days earlier.
**Earth (or Venus) equator, Equinox of date.

More detailed characteristics of the 3-burn transfer for a span of launch dates is shown in Table 2 for the 2007 mission. Here, the Ariane launch dates, in 10-day increments are shown in column 1. Column 2 shows the wait time (discussed above) before the 3-burn initiation. The first burn of MEGA then injects into the high ellipse with the apogee radius shown, but solar perturbations will alter these values. The second burn at apogee to target the lunar flyby has the ΔV shown for each launch date, with the resulting ecliptic inclination of the apogee-to-Moon trajectory segment. The third burn values and the sum of the three burns are shown in the last two columns.

Table 2. MEGA 3-Burn Summary for Venus 2007 Type 2 Trajectory*

Launch Date (2006)	Wait Time (days)	Apogee Radius (10^6 km)	Apogee ΔV (m/s)	Apogee Incl. (deg)	Lunar** Flyby (km)	Escape ΔV (m/s)	Total ΔV (m/s)
July 13	260	1.394	308	23.5	7270	415	1470
Aug. 2	240	1.421	122	21.8	9160	407	1270
Aug. 22	220	1.292	61	16.1	13040	398	1200
Sept. 11	200	1.148	208	19.3	33500	376	1325
Oct. 1	180	1.055	235	97.6	24750	360	1330
Oct. 21	160	1.029	228	143.5	10950	342	1305
Nov. 10	140	1.056	193	150.7	6137	326	1255
Nov. 30	120	1.133	172	153.7	3710	308	1220
Dec. 20	100	1.247	188	155.9	2320	288	1220

*Earth Escape: May 30, 2007 (C_3 = 7.5 km^2/s^2)
Venus Arrival: Nov. 5, 2007
**The lunar flyby date is May 27, 2007.

THE DOUBLE MINIMUM FOR ΔV

Several conclusions may be drawn concerning the general behavior of the MEGA trajectories as the Ariane launch date progresses, and the noon GTO apogee turns to follow the sun. Since the Moon's position is fixed, the trajectory from apogee to the Moon changes from being direct to being retrograde, as shown by the inclination value in column 5. Also, the turning angle at the Moon changes from a sharp direct flyby to practically no turn to a sharp retrograde flyby (column 6). The sharp direct is an energy reducing flyby, thus requiring a larger escape burn at Earth, whereas the retrograde flyby adds energy and reduces the escape burn requirement. This gravity assist by the Moon for late launches actually produces a second minimum which shows up in all of the velocity plots (Figures 4-7).

These characteristics and solar perturbation effects may be seen in Figures 8 and 9 which give polar and 3-D plots for the August 22 and the November 30 Ariane launch dates. Figure 9, in particular illustrates the effectiveness of the MEGA process in reducing a high GTO inclination using the lunar flyby.

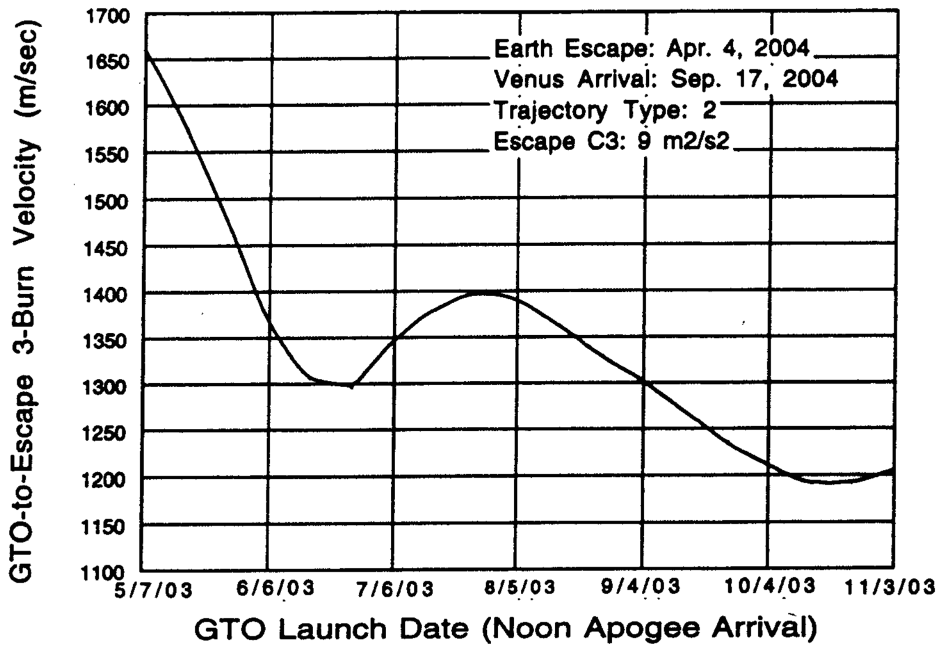


Fig. 4 MEGA Velocity Requirements for Venus 2004

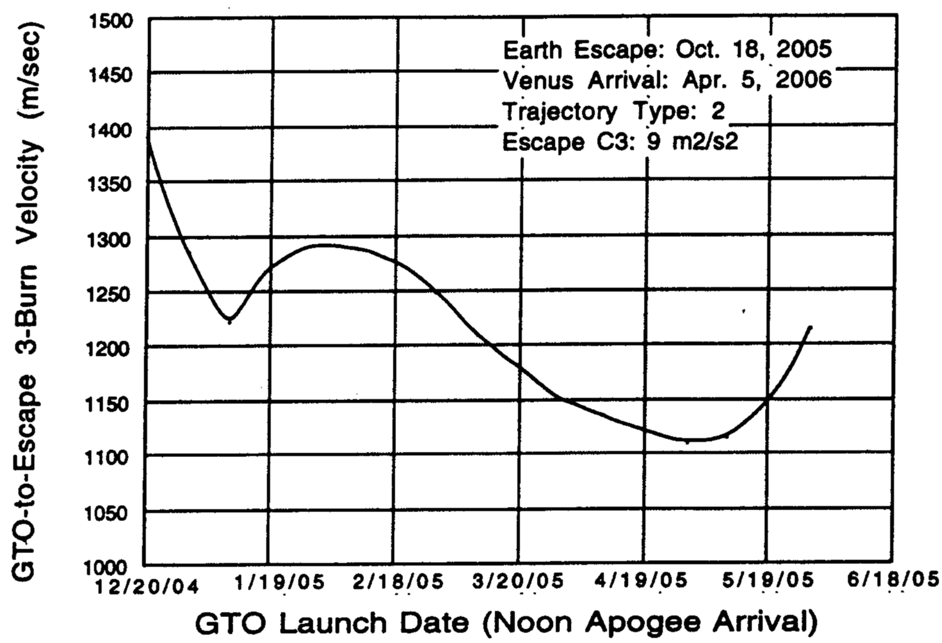


Fig. 5 MEGA Velocity Requirements for Venus 2005 - Early

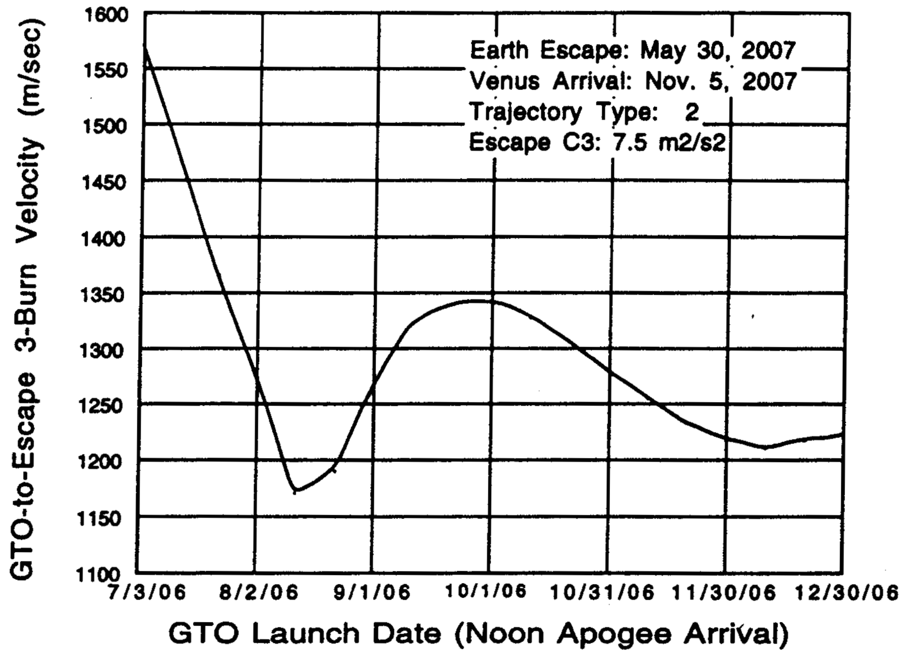


Fig. 6 MEGA Velocity Requirements for Venus 2007

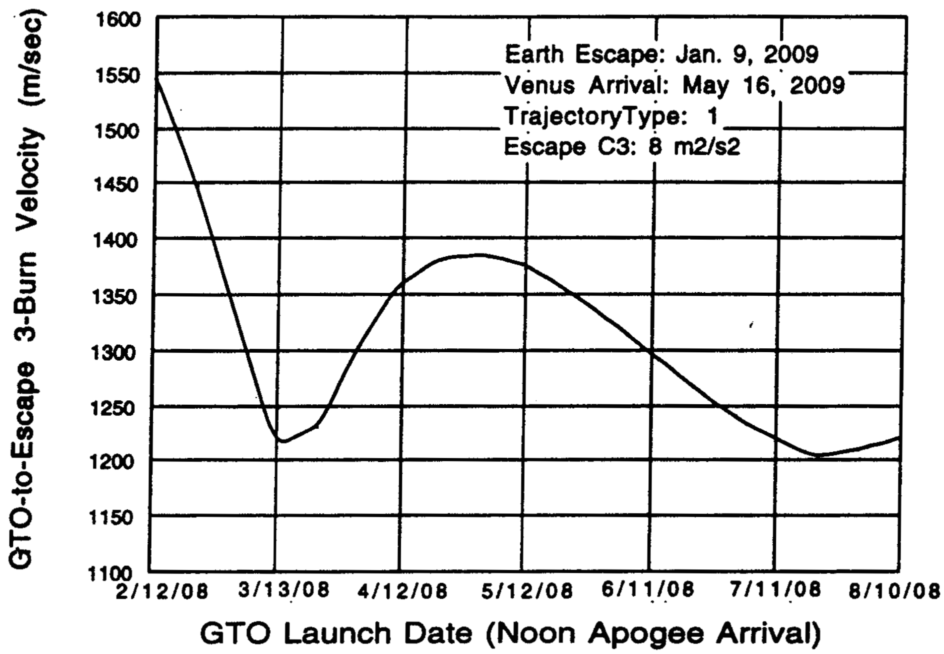
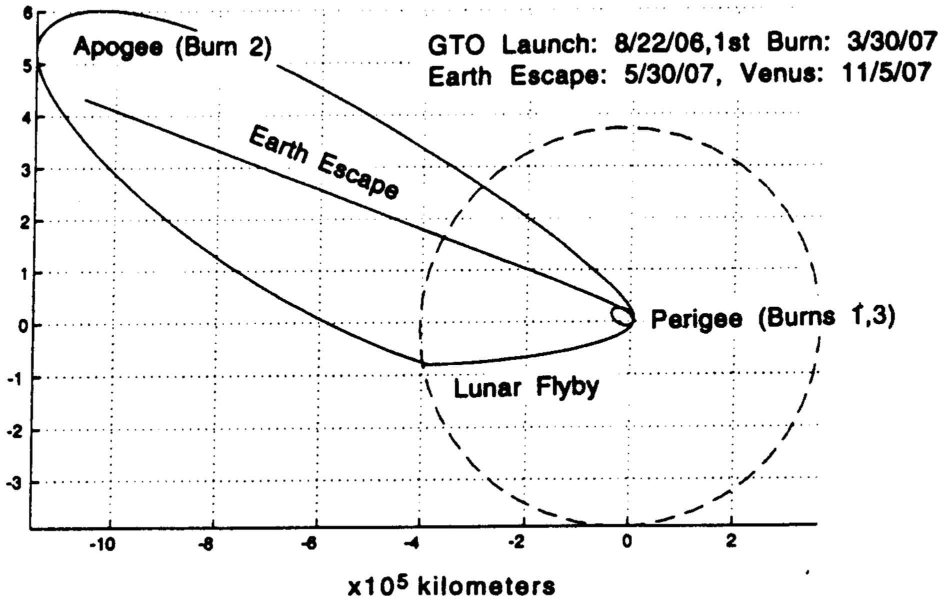


Fig. 7 MEGA Velocity Requirements for Venus 2009



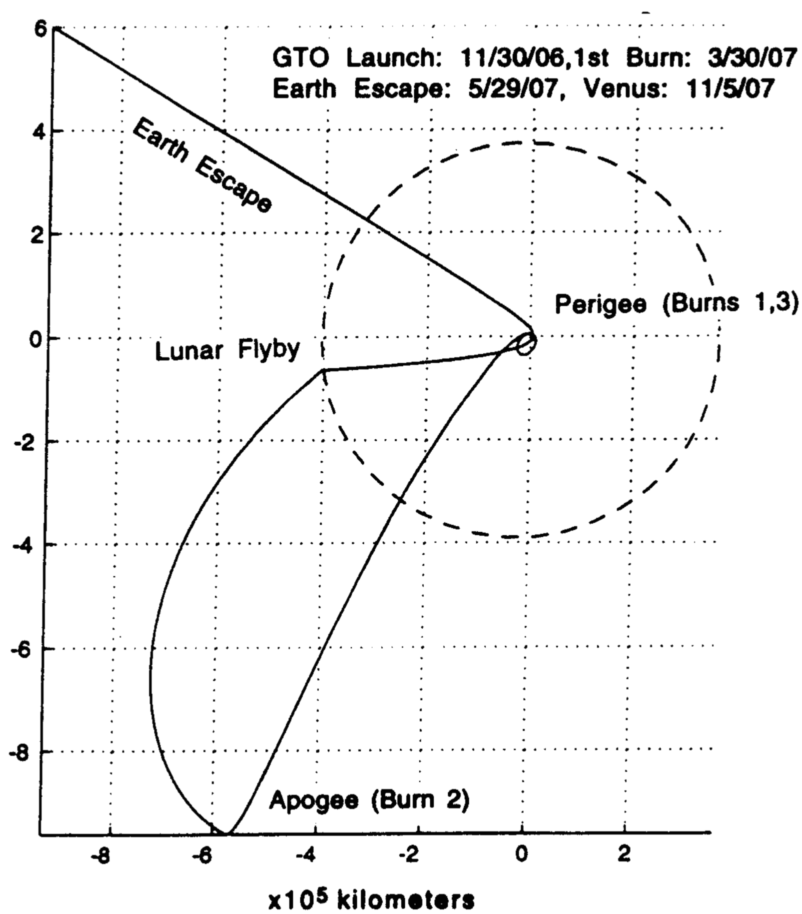
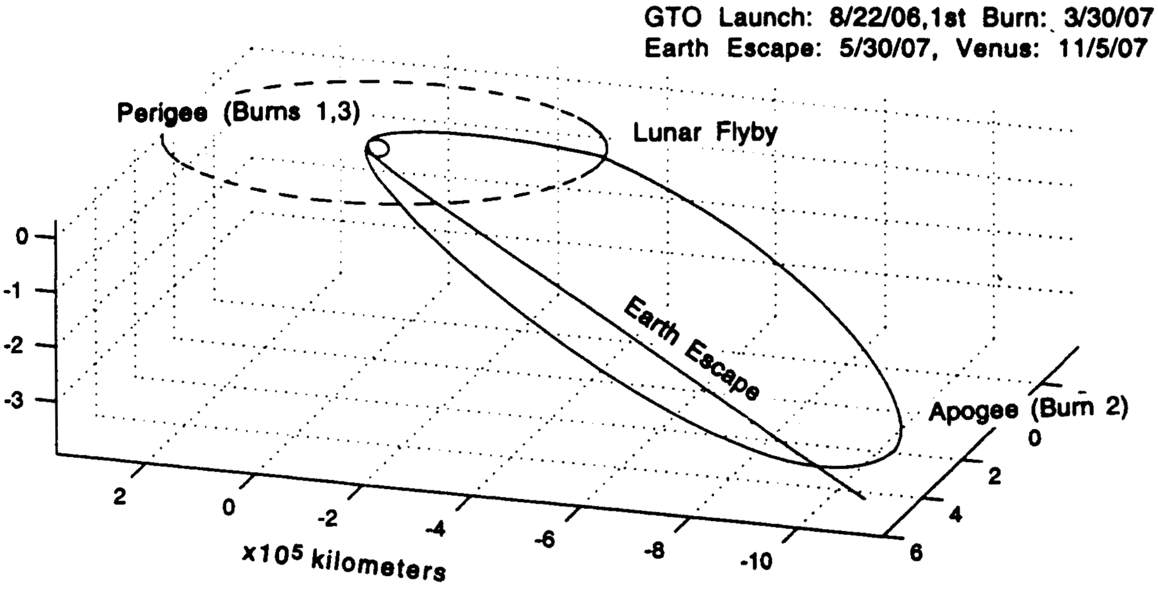


Fig. 8 MEGA Trajectory Plots for Venus 2007 (Polar Views)



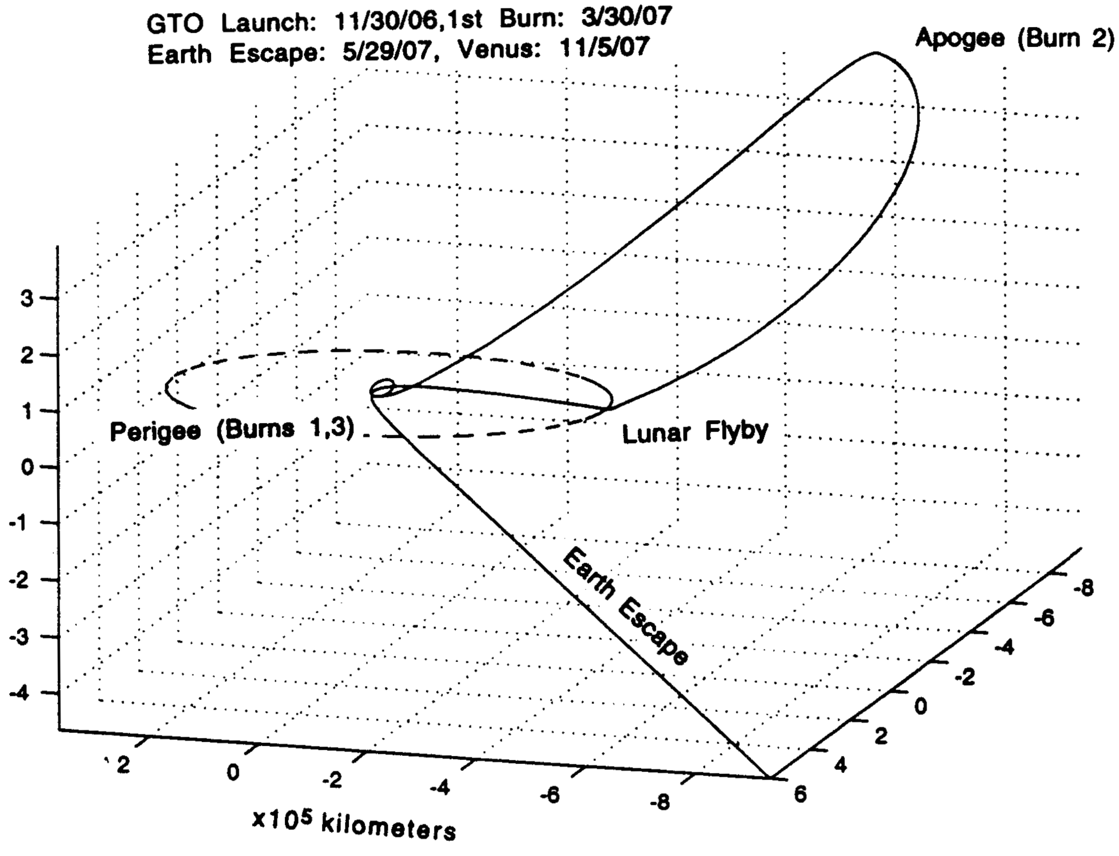


Fig. 9 MEGA Trajectory Plots for Venus 2007 (3-D Views)

OPTIMIZATION OF THE MEGA TRAJECTORY

The MEGA process described here restricts burns one and three to be executed at perigee. The first burn is simply performed at the perigee of the GTO (or higher orbit) available. The high apogee burn is constrained to a zero flight path angle before and after the burn. These are reasonable constraints, and do yield near-optimum results in the solution region of interest, i.e., low ΔV values for the apogee burn.

However, in a global sense, the problem requires optimization. The burns should be free to be performed away from perigee; the apogee burn should not have the flight path angle constraint, and the high ellipse period should not be fixed at 60 days. After all, the real concern is to get the lowest total ΔV to minimize the propellant burned. The period optimization has not been done here, but has been performed for the Mars study of Reference 5. Of course, the greater the period chosen, the higher the apogee distance, which results in a larger solar influence, which may be good or bad for the optimization. A suitable method of optimization is currently being investigated for MEGA type trajectories.

BEYOND VENUS VIA GRAVITY ASSIST

Venus and Mars are the nearest planets to Earth, and the easiest to get to. Venus, being almost as large as the Earth, is also effective as a gravity assist planet to get to Mercury, the asteroids and comets, and to the giant planets and their satellites. Venus gravity assist has been used on the 1973 Mariner Venus Mercury (MVM) mission and more recently on the 1998 Cassini mission to Saturn. Many trajectory studies confirm its usefulness in enabling difficult missions.

Since the MEGA approach can be used for any launch date (within the lunar flyby constraints) and for any Venus arrival date, it may be used as a launch technique for secondary payloads for these Venus gravity assist (VGA) missions. It may be used even more effectively on multiple gravity assist, such as Venus-Earth (VEGA), or Venus-Earth-Earth (VEEGA) missions.

A discussion of the MEGA application to other missions is given in Reference 6. Here, only two examples are given to emphasize the importance of VGA in reaching out to other bodies of the solar system. The first is a VGA mission to Mercury with a 2007 Earth departure. Here, the flight time would be 6 months and require a large maneuver (550 m/s) about 40 days before arriving at Mercury (Figure 10).

The second is a mission to the well known main belt asteroid Vesta (Figure 11). There is no deep space maneuver required, but the escape energy is higher (C_3 = 11.2 m^2/s^2) which would add about 150 m/s to the values shown in Figure 6.

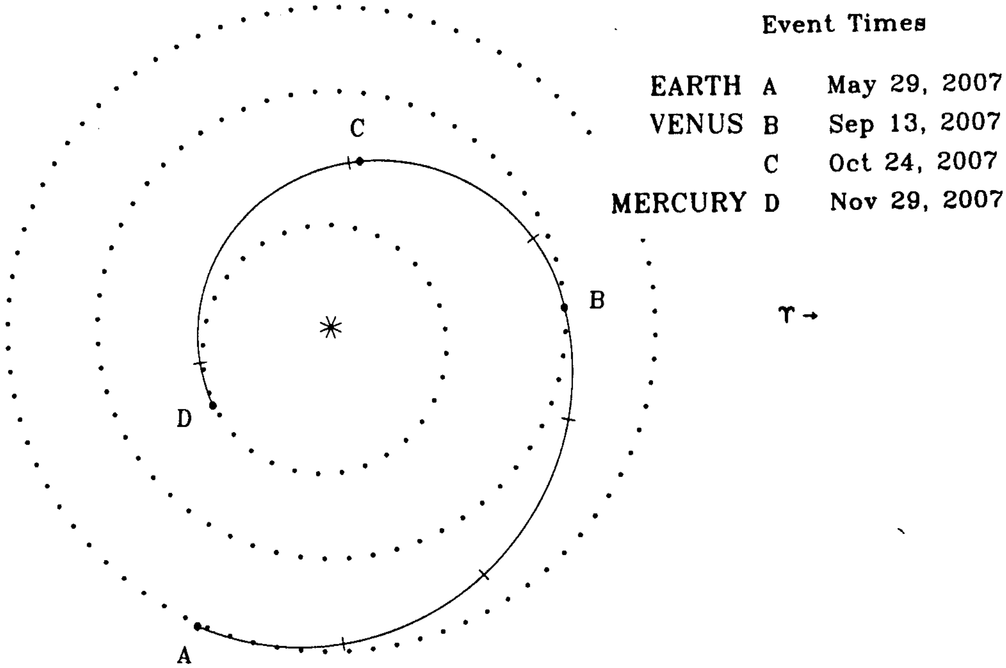


Fig. 10 2007 Venus Gravity Assist (VGA) to Mercury

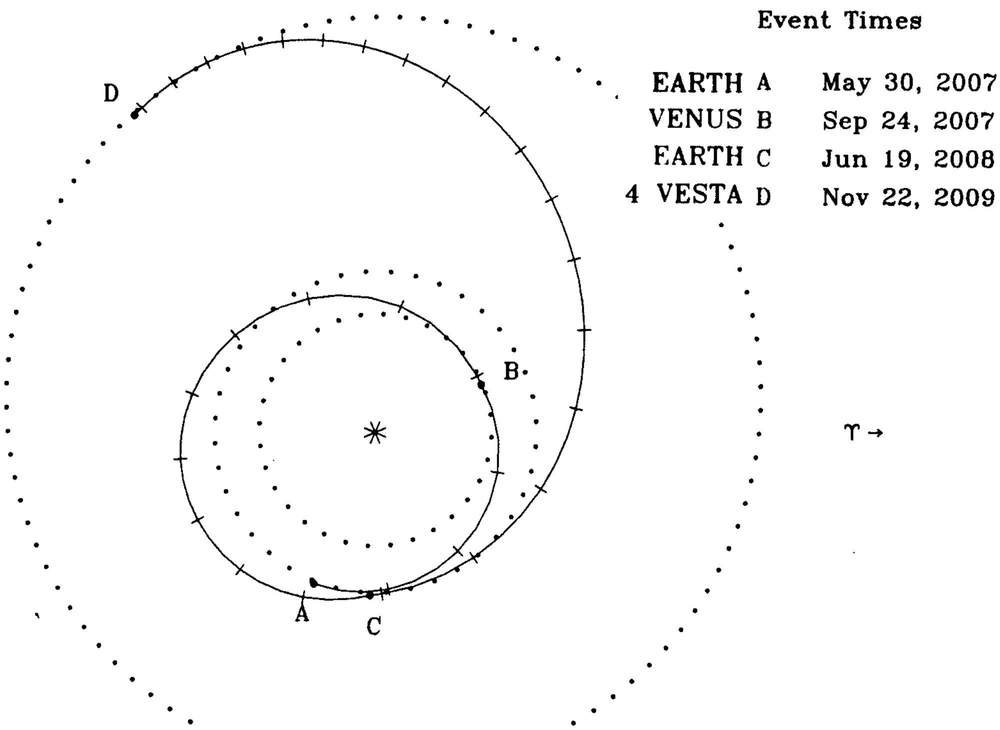


Fig. 11 2007 Venus Earth Gravity Assist (VEGA) to 4 Vesta

SUMMARY

Technology is moving rapidly in the development of smaller and smarter spacecraft which can perform planetary missions. The MEGA process described here provides a means for delivering these spacecraft to Venus as secondary payloads on the Ariane 5 launch vehicle, maintaining a balance between spacecraft, mission, and launch costs. An important element in the feasibility and acceptability of this mode of transportation is the ability to be independent of the day and time of the GTO launch, which is the primary emphasis of this paper.

It has been shown that a five month or more launch period can be developed for Venus missions, with significant latitude in the time of launch during the day. This has been applied to Venus launch opportunities of 2004 to 2009. The long launch period comes about because of a double ΔV minimum, one of which is due to a significant lunar gravity assist, reducing the third or Earth escape burn. Global optimization of the trajectories generated in this study should reduce these ΔV's further. Finally, it is anticipated that additional MEGA modes will be discovered which will expand the launch space, and add greater flexibility in the use of the Ariane ASAP launch capability, and other launch vehicles which may offer to provide secondary payloads for the coming generation of small spacecraft.

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